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(56) Documents Cited
GB 2149022 A **DD 000204969 A**
JP 580093903 A **SU 001517417 A1**

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INT CL^7 **F01D 5/18 9/02 9/04 17/12 17/14 17/16 , F02C 9/16 9/20**
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(54) Abstract Title

Gas turbine nozzle guide vane having a thermally distortable trailing edge portion

(57) To alter the capacity of a turbine of a gas turbine engine by varying the throat area between adjacent nozzle guide vanes, a trailing edge portion of a nozzle guide vane aerofoil suction surface is thermally distorted. The distortion may be controlled by modulating the amount of cooling air flowing through the vane. Cooling air is supplied to at least two cavities 38,40 of the vane, the air to a trailing edge cavity 40 cooling the trailing edge distortable portion. The pressure side of the guide vane aerofoil may resist thermal distortion, while the trailing edge may be bowed (fig 3) and deflected by radial compression forces. Alternatively the trailing edge portion may be a bimetallic strip (56, fig 5).

Fig.3b.

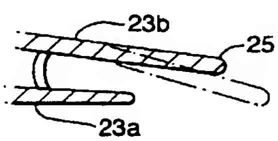
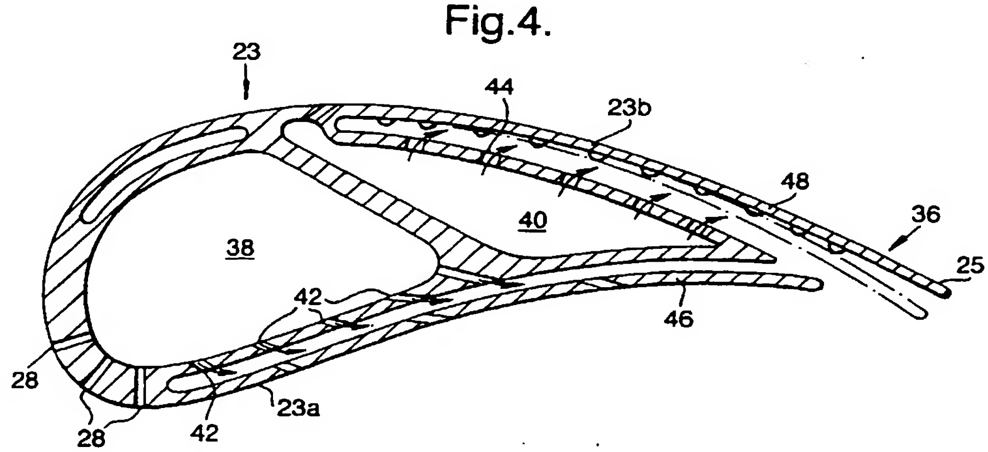


Fig.4.



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At least one drawing originally filed was informal and the print reproduced here is taken from a later filed formal copy.

Fig.1.

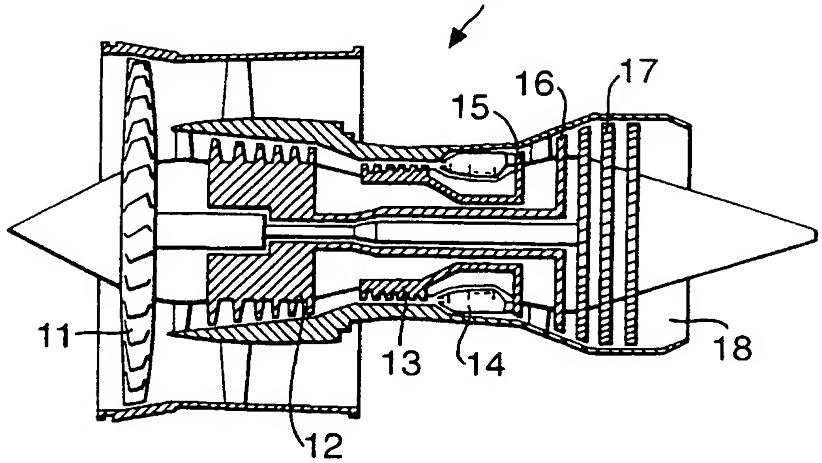


Fig.2.

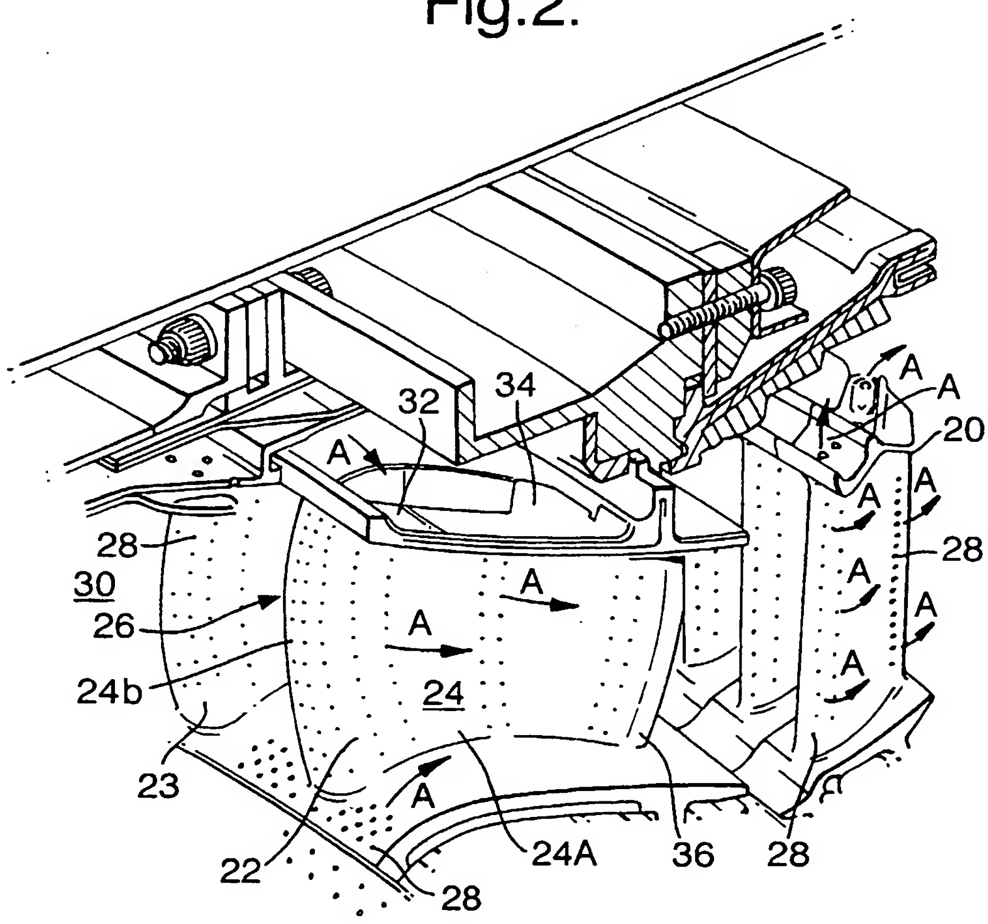


Fig.3.

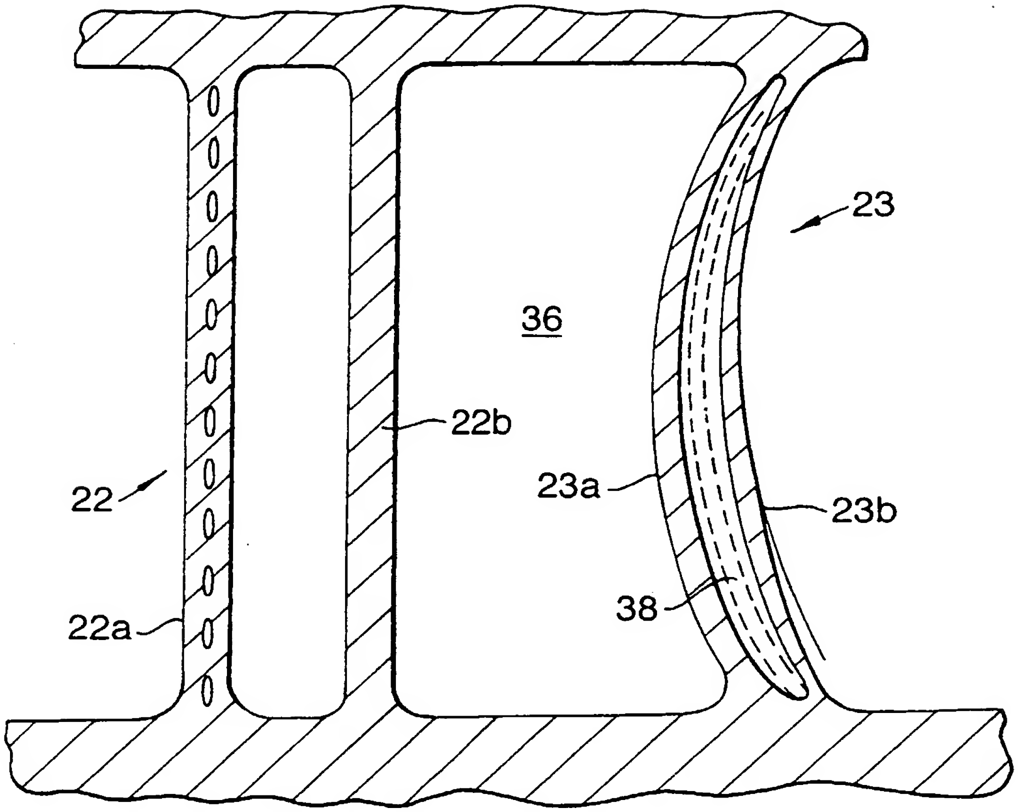


Fig.3b.

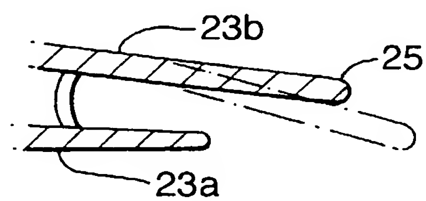


Fig.3a.

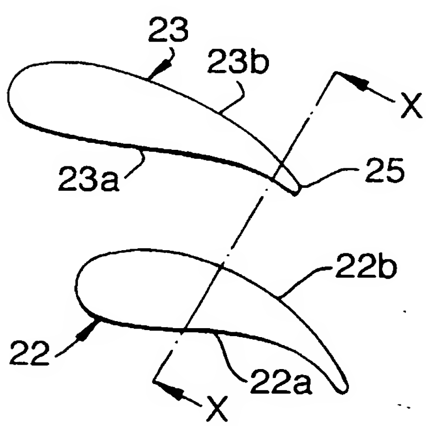


Fig.5.

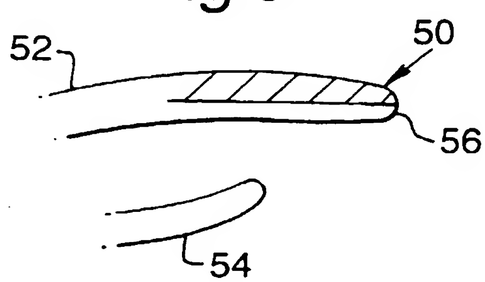
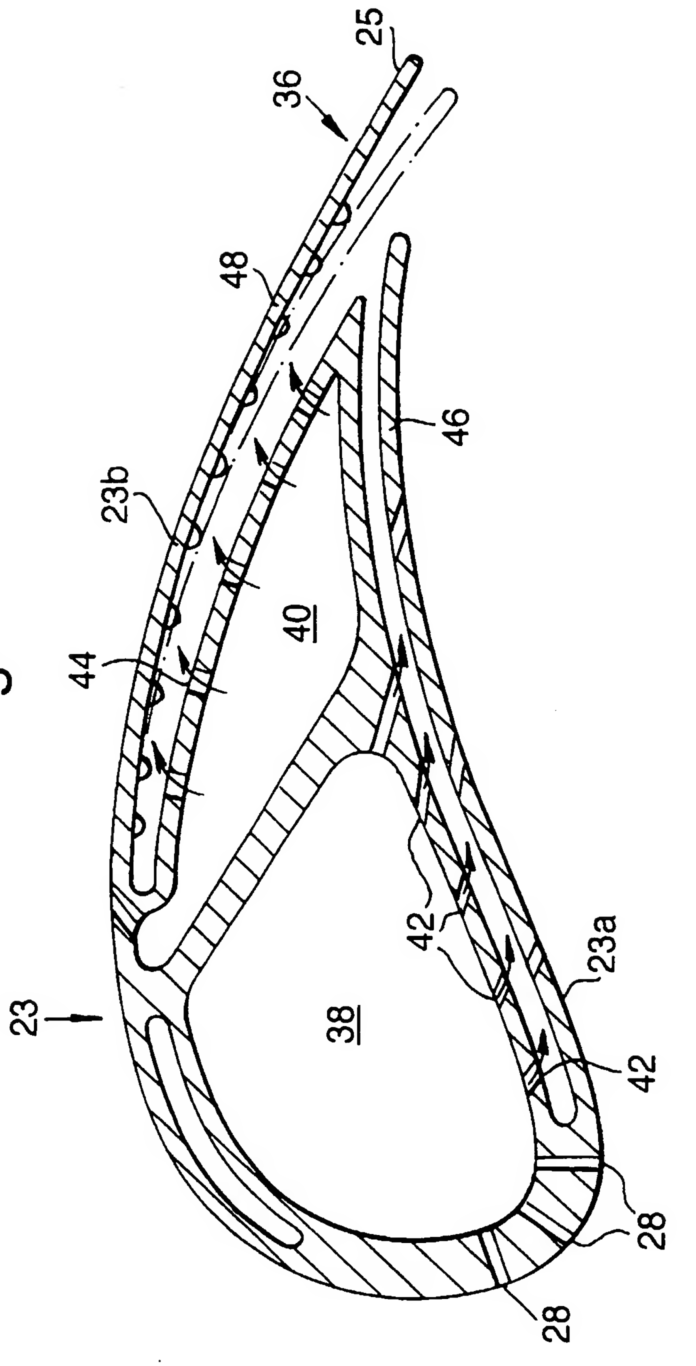


Fig.4.



GAS TURBINE ENGINE SYSTEM

This invention relates to a gas turbine engine. More particularly this invention relates to the turbine and nozzle guide vanes of a gas turbine engine and a method of varying the capacity of the turbine.

The turbine of a gas turbine engine comprises several stages, each stage employing one row of stationary nozzle guide vanes and one row of rotating turbine blades. The turbine depends for its operation on the transfer of energy between the combustion gases and the turbine whilst minimising thermodynamic and aerodynamic losses. When the gas is expanded by the combustion process it is forced into the nozzles of the turbine where, because of their convergent shape, it is accelerated. The nozzle guide vanes provide a spin or 'whirl' in the direction of rotation of the turbine blades.

The performance of an engine is dictated to a large extent by the type of operation for which the engine is designed. The thrust developed by the engine is dependent on the mass of air entering the engine and the acceleration imparted to it during the engine cycle. However it is influenced by such variables as the forward speed of the aircraft, altitude and climatic conditions. For example a large thrust is required during take-off as compared to that required at altitude.

It is desirable for a gas turbine engine to have a variable cycle to accommodate the different requirements for thrust throughout a flight. One method of achieving this would be to vary the throat area between the high pressure nozzle guide vanes of the turbine, hence reducing and increasing the airflow through the nozzle according to thrust requirements.

However a variable flow turbine nozzle is not straightforward to design. The turbine operates under extremely high temperatures and any device designed to move the nozzle guide vanes must also be able to withstand such thermal loads. It is also essential to provide the turbine nozzle guide vanes with cooling holes or slots to provide both impingement cooling, internal cooling and film cooling. The vanes are hollow and the internal cavities receive cooling air, usually from the compressor, which is exhausted through slots or holes.

Previous proposals for achieving high pressure nozzle guide vane modulation have been complicated, expensive or unreliable. Prior patent No US 5931636 discloses one such proposal whereby the suction side of each vane is rotatable so as to vary the throat area. This proposal involves the use of a complex cam mechanism adding weight to the turbine.

It is an aim of this invention therefore to provide a variable area turbine nozzle, which attempts to alleviate the aforementioned problems.

According to the invention there is provided a gas turbine engine comprising an array of nozzle guide vanes each vane comprising an aerofoil member having a pressure surface, a suction surface, a trailing edge portion and at least two internal cavities for receiving cooling air and exhaust means for exhausting cooling air from said cavities, a first cavity being formed within the trailing edge region of said aerofoil and control means being provided to control the amount of cooling air directed through said first cavity , said suction surface being at least partially flexible at the trailing edge portion adjacent said first cavity such that during use of the engine this portion of the suction surface distorts and

moves relative to an adjacent aerofoil member when the supply of cooling air into said first cavity is modulated.

Also according to the present invention there is provided a method of controlling the capacity of a turbine for a gas turbine engine comprising the step of controlling the amount of cooling air through at least one nozzle guide vane aerofoil member so as to thermally distort its suction surface trailing edge portion such that the throat area between at least two adjacent nozzle guide vane aerofoils is altered.

An embodiment of the invention will now be described with respect to the accompanying drawings in which:

Figure 1 is a schematic section view of a ducted gas turbine engine incorporating a number of turbine nozzle guide vanes in accordance with the present invention.

Figure 2 is a view of a nozzle guide vane and turbine blade arrangement of a gas turbine engine in accordance with an embodiment of the present invention.

Figure 3 is a section through X-X of figure 3a showing the throat plane between two adjacent vanes according to the present invention.

Figure 3a is a top view of the nozzle guide vanes of figure 3.

Figure 3b is an enlarged section view of the trailing edge of one of the vanes of figures 3 and 3a.

Figure 4 is a section view of a turbine blade in accordance with a further embodiment of the invention.

Figure 5 is an enlarged view of the trailing edge portion of a nozzle guide vane according to a further embodiment of the invention.

With reference to figure 1, a ducted gas turbine engine shown at 10 is of a generally conventional configuration. It comprises in axial flow series a fan 11, intermediate pressure compressor 12, high pressure

compressor 13, combustion equipment 14. The turbine equipment comprises high, intermediate and low pressure turbines 15, 16 and 17 respectively and an exhaust nozzle 18. Air is accelerated by the fan 11 to produce two flows of air, the larger of which is exhausted from the engine 10 to provide propulsive thrust. The smaller flow of air is directed into the intermediate pressure compressor 12 where it is compressed and then directed into the high pressure compressor where further compression takes place. The compressed air is then mixed with the fuel in the combustion equipment 14 and the mixture combusted. The resultant combustion products then expand through the high, intermediate and low pressure turbines 15, 16 and 17 respectively before being exhausted to atmosphere through the exhaust nozzle 18 to provide additional propulsive thrust.

Now referring to figure 2 a high pressure turbine 15 for a gas turbine engine is shown in a partial broken away view. The turbine includes an annular array of similar radially extending air cooled aerofoil turbine blades 20 located upstream of an annular array aerofoil nozzle guide vanes, two of which 22, 23 are shown in figure 2. The turbine is provided with several more axially extending alternate annular arrays of nozzle guide vanes and turbine blades, however these are not shown in figure 2 for reasons of clarity.

The nozzle guide vanes 22, 23 each comprise an aerofoil portion 24 with the passage between adjacent vanes forming a convergent duct 26. The turbine blades 20 also comprise an aerofoil portion 25. The vanes are located in the turbine casing in a manner that allows for expansion of the hot air from the combustion chamber. Both the nozzle guide vanes 22, 23 and turbine blades 20 are cooled by passing compressor delivery air through them. Arrows A

indicate this cooling air. Cooling holes 28 provide both film cooling and impingement cooling of the nozzle guide vanes and turbine blades.

In operation hot gases flow through the annular gas passage 30. These hot gases act upon the aerofoil portions of the turbine blades 20 to provide rotation of the turbine disc (not shown) upon which the blades are mounted. The gases are extremely hot and internal cooling of the vanes 22, 23 and the blades 20 is necessary. Both the vanes 22, 23 and the blades 20 are hollow in order to achieve this and in the case of vanes 22, 23 cooling air derived from the compressor is directed into their radially outer extents through apertures 32 formed within their radially outer platforms 34. The air then flows through the vanes 22 to exhaust therefrom through a large number of cooling holes 28 provided in the aerofoil portion 24 into the gas stream flowing through the annular gas passage 30.

The cooling required at cruise conditions is less than required during take-off, for example. As such part of the vane 22 may be allowed to increase its temperature at cruise without causing failure.

Now referring to figures 3, 3a and 3b, two nozzle guide vanes 22, 23 are shown adjacent each other and both comprise a pressure surface 22a, 23a and a pressure surface 23b, 23b. Figure 3 is a section through the vanes at X-X of figure 3a, and as such figure 3 shows a section taken through the trailing edge region of vane 23 and a section through the middle portion of vane 22. The view in figure 3b shows an enlarged view of the trailing edge region of the vane 23. To reduce the throat area 36 between the nozzle guide vanes 22, 23 the suction surface 23b of nozzle guide vane 23 is thermally distorted. This distortion is shown by the dotted lines in figure 3 and 3b. The trailing edge portion 25 of the suction surface 23b moves toward the

suction surface 22b and hence reduces the throat area 36.This distortion may be affected across the whole nozzle by thermally distorting alternate vane aerofoils. Thus the capacity of the turbine is reduced as the throat areas are reduced.

The inner portion of nozzle guide vane 23 is divided into two cavities, one 40 within the trailing edge 25 and one 38 within the leading edge of the aerofoil. The cavities cannot be seen in figure 3 but are shown in figure 4. Thermal distortion of the vane 23 is provided by modulating the amount of cooling air passing through into cavity 40 within the vane 23. This modulation is achieved by a valve arrangement and appropriate control system located further upstream which controls and modulates the amount of cooling compressor air made available to this cavity 40. Thus when the capacity of the turbine is required to be reduced the cooling compressor airflow to the cavity 40 may be reduced or restricted causing an increase in temperature and hence thermal distortion of the suction side of the vane 23.

In the embodiment shown in figure 3 the suction surface 23b of vane 23 is manufactured to be 'flexible' or capable of thermal distortion whilst the pressure surface 23a is manufactured to be stiff and resist thermal distortion. Through modulation of the cooling air to the cavity 40 of vane 23, the temperature of the trailing edge portion of the suction surface 23b is raised substantially, relative to the remainder of the vane aerofoil. This temperature difference may also be increased if the rear portion is 'over-cooled' at take off hence later allowing a substantial increase in temperature and without compromising the thermal limits of the vane material.

The trailing edge 25 of a nozzle guide vane is normally slightly bowed towards a neighbouring vane suction

surface as shown in figure 3. The trailing edge region 25 is put into radial compression by the restriction of cooling airflow to the rear portion of the vane 23 causing it to deflect toward the neighbouring vane suction surface 22b.

The present invention therefore uses the thermal mismatch between the front and rear portions of the aerofoil to vary the high pressure nozzle guide vane throat area and hence vary the turbine capacity.

The structure of a nozzle guide vane 22 according to another embodiment of the present invention can be seen more clearly by referring to figure 4. The vane 23 comprises a suction surface 23b and a pressure surface 23a. These surfaces 23a, 23b converge at the trailing edge portion 25 of the vane 23. The suction surface 23b comprises an outer flap 48 which extends from the substantially central portion of the surface 23.

The vane 23 is divided into two cavities 38, 40 both suitable for receiving cooling air form the compressor. The cooling air from the first, leading edge cavity is exhausted through holes 28 and 42 and the air form the rear cavity 490 is exhausted through holes 44. The direction of airflow is indicated by the arrows. The pressure surface 23a and suction surface 23b of the vane 23 both comprise and outer skin 46, 48. The suction surface skin or flap 48 is flexible so as to be capable of thermal distortion (indicated by the dotted lines). The pressure surface cooling as arranged such that its temperature is not greatly altered by the cooling air modulation. The platforms on which the vanes 22, 23 are mounted, are stiff enough to constrain the ends of the flexible plate to follow the main aerofoil, thus forcing the suction side into compression when it is warmed relative to the main aerofoil.

In a further embodiment of the invention shown in figure 5 it is also envisaged that the overhang portion of a nozzle guide vane, that being the extent to which the suction surface 52 extends beyond the pressure surface 54 at the trailing edge, could be manufactured as a bimetallic strip 56. In this case the characteristics of a bimetallic strip would produce the necessary thermal distortion of the suction surface 52. If the cooling air is not varied the vane temperature at cruise conditions is substantially lower than at take-off. The bimetallic strip may use this temperature change to deflect the trailing edge toward the neighbouring vane suction side at cruise as shown in figure 5b, thus providing the required behaviour without the need for a complicated cooling air modulation system. The required deflection is smaller than that shown in figure 5b which is exaggerated for explanatory purposes.

CLAIMS

1. A gas turbine engine comprising a turbine said turbine comprising an array of nozzle guide vanes each vane comprising an aerofoil member having a pressure surface, a suction surface, a trailing edge portion and at least two internal cavities for receiving cooling air and exhaust means for exhausting cooling air from said cavities, a first cavity being formed within the trailing edge region of said aerofoil and control means being provided to control the amount of cooling air directed through said first cavity , said suction surface being at least partially flexible at the trailing edge portion adjacent said first cavity such that during use of the engine this portion of the suction surface distorts and moves relative to an adjacent aerofoil member when the supply of cooling air into said first cavity is modulated.

2. A turbine as claimed in claim 1 wherein at least the trailing edge portion of said aerofoil suction surface comprises an external flap, said flap being free at its trailing edge and being capable of movement under temperature changes.

3. A turbine as claimed in any one of the preceding claims wherein the second cavity is formed within the leading edge portion of the aerofoil.

4. A turbine as claimed in any one of the preceding claims wherein said turbine comprises the high pressure turbine of a gas turbine engine.

5. A turbine as claimed in claim 1 wherein the trailing edge comprises a sandwich of two parts, one part having a higher thermal expansion coefficient than the other part such that the trailing edge deflects under the application of heat.

6. A method of controlling the capacity of a turbine for a gas turbine engine comprising the step of controlling the amount of cooling air through at least one nozzle guide vane aerofoil member so as to thermally distort its suction surface trailing edge portion such that the throat area between at least two adjacent nozzle guide vane aerofoils is altered.

7. A method according to claim 6 wherein the trailing edge of said suction surface is subjected to compression under the application of heat so as to deflect toward an adjacent aerofoil vane suction surface.

8. A method according to claims 6 to 7 wherein the compression is radial compression and is achieved by reducing the amount of cooling air available to said suction surface trailing edge portion.

9. A method of controlling the capacity of a gas turbine engine turbine substantially as described herein with reference to the accompanying drawings.

10. A turbine for a gas turbine engine substantially as described herein with reference to the accompanying drawings.

The Patent Office

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Application No: GB 0103821.5
Claims searched: 1-8
Examiner: Terence Newhouse
Date of search: 6 June 2001

Patents Act 1977
Search Report under Section 17

Databases searched:

UK Patent Office collections, including GB, EP, WO & US patent specifications, in:

UK Cl (Ed.S): F1V(VCT,VDCC)

Int Cl (Ed.7): F01D 9/02 9/04 17/12 17/14 17/16 5/18; F02C 9/16 9/20

Other: ONLINE: EPODOC, JAPIO, WPI

Documents considered to be relevant:

Category	Identity of document and relevant passage		Relevant to claims
X	GB 2149022 A	(ROLLS), see particularly fig 3 and page 3 lines 38-46	1,6 at least
X	DD 204969 A	(GEBHARDT), see also WPI Abstract Accession No. 1984-089120[15]	6 at least
A	SU 1517417 A1	(KAZAN AVIATION), see also WPI Abstract Accession No. 1993-173678[21]	
X	JP 580093903 A	(HITACHI), see also entry in Patent Abstract of Japan and figs 4 & 5 noting thermally deformable plate 10	1,6 at least

X Document indicating lack of novelty or inventive step
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